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THE LUNAR ORBITER

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THE LUNAR ORBITER

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Introduction

The Lunar Orbiter has as its prime objective the photographic coverage of selected portions of the lunar surface. Although the Ranger Program has provided some extremely valuable high resolution lunar photographs, the coverage is too limited to provide sufficient information for Apollo. The Lunar Orbiter coverage will be secured with sufficient resolution and will cover enough area to allow the selection of landing sites for Apollo, and in conjunction with the Surveyor Spacecraft will provide necessary information for the safe landing of Apollo. The spacecraft has been designed around its photographic payload, and does not carry many subsidiary experiments; however, it will provide information regarding the lunar gravitational field, and a limited amount of information regarding micrometeorites and energetic particles.

The basic concepts described in this paper were initiated by The Boeing Company in response to a request for proposal issued to industry by the NASA. The Boeing Company is carrying out this project, assisted by two major subcontractors, Eastman-Kodak and RCA. The project is being managed by the NASA Langley Research Center.

Mission Plan

It is planned that the first Lunar Orbiter Spacecraft will be launched in 1966, using an Atlas Agena vehicle. The launch technique and procedures (fig. 1) will be very similar to those used for the Ranger Spacecraft, except for the launch windows and the targeting required for proper lunar orbits. These are unique for the Orbiter. Tracking during launch will be accomplished by the Eastern Test Range stations; during subsequent phases of the mission tracking will be by the NASA Deep Space Network receiving stations. Centralized control of the mission will be provided from the NASA Space Flight Operations Facility in Pasadena, California.

A cislunar trajectory which will take from 60 to 90 hours is planned. Figure 2 shows a typical series of maneuvers which will occur at the moon. Although these are shown in the plane of the approach trajectory, there will be the capability of making a plane change during main deboost in order to more quickly cover the prime photographic targets. The approach hyperbola, with closest approach at about 900 Km above the lunar surface is planned so that at main deboost the flight path is parallel to the initial orbit ellipse. Main deboost is adjusted so that the initial orbit has an apolune of 1800 Km and a perilune of about 200 Km. The line of apsides of this orbit is adjusted to secure coverage of the selected target. After a period of tracking by the Deep Space Receiving Stations to determine the orbital parameters, a final orbit adjustment is made by means of a Hohman transfer to provide a periselene of 46 Km above the photographic target. The launch time and orbit transfers must be planned so that when the spacecraft arrives above the target proper lighting is available for photography. Proper lighting conditions are those which provide sufficient light and enough sensitivity to surface slope changes to secure adequate signal-to-noise in the final photograph. Figure 3 shows the lunar photometric function and identifies the region of operation which is employed by the Lunar Orbiter camera. The function ϕ is a measure of the reflectivity of the surface, g is the phase angle (angle between line of sight and sun line), and α is the inclination of the surface normal with respect to the line of sight. The case wherein these angles all lie in one plane is illustrated in figure 3. The plot gives the sensitivity of the photographic system with lighting angle to changes in slope of the surface. For example, at a constant value of $g = 60^\circ$, if the slope changes $+15^\circ$ (away from the sun), the light intensity drops to one-half the intensity at $\alpha = 0$. The low-resolution camera planned for the Lunar Orbiter has a field of view of about $\pm 20^\circ$ and will cover the operating range shown within the shaded area. There is good sensitivity to slopes within this area; however, the absolute intensity of a flat lunar surface may vary by almost 2:1 across the frame.

Figure 4 illustrates how the lighting conditions progress with respect to the lunar surface for the 17° inclination case. Posigrade orbits are shown with the initial ascending nodal point at about 50° W. longitude. An arbitrary target has been selected and is shown at 20° E., 3° N. With no plane change during the orbital maneuvers, an initial waiting time of 4 days is required in order for the descending node portion of the orbit to begin intersecting the region of interest. After an 8-day waiting period, the 54th orbit intercepts the target. The timing of this event is such that there is proper illumination ($g = 60^\circ$). It is also possible on this and other orbits with proper spacecraft attitude to take satisfactory pictures over the range included by the shaded area. Boundaries of the shaded areas are limits set by the range of lighting (g variable from 75° to 50°) which yields a satisfactory picture. It should be noted that as the orbit rotates with respect to the lunar surface at 13° per day and the illumination rotates in the same direction at the rate of 12° per day there is a slight vernier effect which shifts the latitude over which satisfactory pictures can be taken. It may be concluded from this figure that it is possible to photograph on a single flight targets located within the shaded area of figure 11 with a maximum waiting time of about 15 days.

Photographic Coverage

The Spacecraft camera, which will be described later in the paper, contains two f/5.6 lenses, a 24-inch focal length high-resolution lens, and a 3-inch focal length lens used for moderate resolution stereo coverage. Simultaneous exposures are made which have the format shown at the top of figure 5. The high-resolution lens covers, from an altitude of 46 Km, a region 16.4 × 4.1 Km with approximately 1 meter resolution. That area lies, as shown, within the 36-Km × 33-Km area covered by the other lens with a resolution of about 8 meters. Two multiple-exposure coverage modes are also shown in figure 5. An example of one mode is the 4 frame stereo sequence which provides moderate resolution stereo coverage of a 36-Km × 50-Km area and 4 nonstereo high-resolution samples. Another mode is shown by the 14 frame high-resolution sequence, which provides contiguous high-resolution coverage of a 16.4-Km × 64-Km area. On successive orbital passes the width of coverage of any selected mode may be increased with a repeated similar sequence. Approximately 200 feet of film is carried in the camera, providing 190 frames of coverage. This allows 12,000 Km^2 of high-resolution coverage and 100,000 Km^2 of stereo coverage. In the 200-foot roll of 70-mm film, approximately one million pictures of quality similar to commercial TV are stored. The high-resolution pictures alone would provide sufficient ground coverage to photograph the entire width of the United States in a swath about 1 mile wide.

Spacecraft Configuration

Figure 6 is a drawing of the 850-pound spacecraft configuration which shows that most of the major spacecraft components and systems are attached to a single equipment mounting plate, with the exception of the rocket engine and tanks. In the flight configuration all of the main spacecraft structure above the equipment mounting deck is covered with a highly reflective shroud of aluminum coated mylar. During launch, the solar panels are folded under the spacecraft base and the antennas held against the side of the structure. The spacecraft is $5\frac{1}{2}$ feet high and 5 feet in diameter, excluding the solar panels and antennas. When deployed, the span along the antenna booms is $18\frac{1}{2}$ feet and about 12 feet across the solar panels. A photograph of the Lunar Orbiter mockup is shown in figure 7. Major elements of the spacecraft are described below.

Spacecraft Structure

The spacecraft consists of a main Equipment Mounting Deck and an upper module supported by trusses and an arch. The module supports the gimballed velocity control engine and tanks. The upper module may be removed as an assembly for engine testing. It also carries, directly under the engine, the high pressure N_2 tank, which provides pressurization for the engine feed system and the attitude control thrusters.

Programmer

The spacecraft programmer is essentially a digital data processing system, consisting of registers, memory, clock, comparators, adders, etc. The unit weighs 16 pounds, uses integrated circuitry, and includes a random access memory with a capacity of 128 21-bit words. It will be used to control about 65 functions within the spacecraft.

As figure 8 illustrates, inputs to the programmer are provided from the earth through the spacecraft communications system. These command inputs are either in the form of a direction to be stored and acted upon at a designated later time or a "real-time" direction to be executed immediately.

Because a large number of the stored commands will involve sequences that will be used repeatedly during the mission, those sequences are permanently incorporated as subroutines in the programmer. The command from the earth, therefore, needs only to designate which sequence is desired, indicate the time at which the sequence is to be initiated and, where required, the "magnitude" for each step of the sequence.

A command to change spacecraft attitude is an example where the magnitude of the desired change about the pitch, yaw, and roll axes is inserted into the sequence. The programmer, however, does not function in an open-loop manner for this type command. When it calls for a particular change in pitch attitude, it will not step to the next function in the sequence until the pitch change is verified by integration of the measured pitch axis angular velocity. If that attitude change was for the purpose, for example, of placing the spacecraft in the proper orientation for deboost into lunar orbit, the programmer would then fire the retro engine, compare the measured change in velocity with that called for by earth-command, and shut the engine off when the desired velocity change had been accomplished. After that, the programmer would initiate the "reverse attitude maneuver" sequence to return the spacecraft to its original orientation, and then the sequence to "acquire sun and Canopus."

Power System

The spacecraft power system is a conventional solar array-storage battery type with appropriate provisions for charge control and voltage regulation. The battery will be used to supply the spacecraft power requirements during the launch phase prior to solar array deployment and during these periods of the lunar orbit when the spacecraft is in the moon's shadow. At all other times the solar array will supply the spacecraft demands, including recharging of the battery.

When in full illumination, the 10,856 solar cells which comprise the array will have a maximum power output of about 375 watts. The "nighttime" capability of the system is represented by the 12-ampere-hour capacity of the nickel-cadmium battery. The output voltage of the system can vary from a minimum of 22 volts when the loads are being supplied by the battery to a maximum of 31 volts when the array is in operation. A shunt regulator is used to prevent the voltage from exceeding 31 volts.

Attitude System

The attitude system block diagram is shown in figure 9 for the yaw axis. A similar system is employed using a Canopus star tracker for roll control. The operational modes for this system are:

(a) Celestial hold - In this mode the basic references are the sun and Canopus with the gyros operating as rate sensors. This mode is employed during normal cruise and is used as a reference for initiation of all attitude changes.

(b) Inertial hold - In this mode the basic references are three gyros operating as attitude angle sensors. This mode is used in any axis where the celestial reference is occulted, during engine firing, and during part of the photographic hold mode.

(c) Slew command - In this mode one axis at a time is commanded to acquire a preset angular rate. Two gyros are in inertial hold and the command axis gyro is in the rate mode. The command axis gyro output is mixed with the slew signal and when a match is secured the jet ceases to fire.

(d) Engine on, inertial hold - This mode is similar to mode (b), except that the velocity control engine itself is used to maintain spacecraft orientation. This is accomplished by gimballing the engine.

(e) Photographic hold - In this mode the pitch and roll axes are in inertial hold. The yaw axis is set to zero yaw (to avoid photographic smear) by nulling the output of a V/H crab angle sensor and it is then returned to inertial hold for the duration of the photographic portion of the orbit.

All of the mode selections and timing and comparison of commanded and actual angle changes are made within the spacecraft programmer, as described previously.

Velocity System

The velocity control engine is a 100-pound thrust Marquardt (MA 109) bipropellant engine. Nitrogen Tetroxide and Aerozine 50 are used, pressure fed by nitrogen gas at about 200 psi pressure. With the planned tank size, a total ΔV capability of about 3280 feet per second will be available for midcourse corrections and lunar orbit adjustments. Control over velocity changes is achieved by using a precision linear accelerometer to integrate velocity changes and command engine shutdown at the appropriate time.

Camera System

Figure 10 is a schematic of the camera system. Two lenses are employed, a 24-inch focal length, f/5.6 for high-resolution pictures and a 3-inch, f/5.6 lens for simultaneous, nested, low-resolution pictures. The film used is a 70-mm Kodak SO 243 aerial film which is preprinted along one edge with grey scales, resolution bars, and other pertinent information. The 24-inch lens produces a frame approximately 55 by 219 mm (corresponding to 4.1 by 16.4 km on the lunar surface from an altitude of 46 Km). The 3-inch lens produces a 55- by 60-mm frame which corresponds to a 33- by 36-Km square from the same altitude. After exposure, the film is stored on loopers. The film is then passed through a Bi-Mat Processor at a speed of about 77 mm/minute. Enough looper storage is provided to store all of the film taken during one orbit; the processor can process all of the film before the next orbit. The processor presses the SO 243 into contact with Kodak SO 111. These are separated after processing, the film dried and passed through the inactive readout onto a film takeup spool. At any time, the film can be read out by running it through the readout onto the readout looper. The capacity of this looper is about four frames. To provide image motion compensation during exposure, the film platens are moved in the flight direction at a speed commanded by a V/H sensor. Transverse image motion compensation is not necessary as the spacecraft yaw angle is commanded to zero in the photographic mode.

Figure 11 is a schematic of the spacecraft readout system. The light source for film scanning is a CBS line scan tube having its phosphor on a revolving drum. The generated spot of light is demagnified to 0.005 mm and produces scanning lines on the film of approximately 2.5-mm length. The demagnifying optical system is moved so that successive line scans are displaced until 60 mm is scanned, then the film is advanced 2.5 mm. The next series of line scans occurs in the opposite direction as shown. Collecting optics lead the transmitted light into a photomultiplier, and the resulting electrical signal is then conditioned for handling by the spacecraft communication system. A separate synch package provides spot sweep voltages to drive the line scan tube and synchronization pulses.

The ground reconstruction equipment located at each of three Deep Space Receiving Stations, accepts the video signal and displays the video data line by line on a kinescope face. The displayed image is recorded on a continuously moving 35-mm film strip. The 35-mm film strips are then delivered to a central facility where a reassembly printer is employed to automatically reassemble the photographs.

Communications System

The spacecraft S-Band communication system, which is shown in block diagram form in figure 12, will provide for the transmission of the spacecraft telemetry and video data, for the reception of commands from the earth and for the signals which permit range and range-rate data to be obtained for use in trajectory and lunar orbit determination. The telemetry and commands will be in a digital form and the video data in analog form.

As shown in figure 12, the spacecraft will employ two antennas, a low-gain one having a reasonably "omnidirectional" pattern and a high-gain antenna with a 10° beamwidth. Command reception, low-power telemetry transmission (0.5 watt), and ranging and range-rate measurements will be accomplished using the low-gain antenna. The 3-foot-diameter parabolic high-gain antenna will be used only during period of photographic data readout for transmission of the composite video and telemetry signal. A 10-watt travelling wave tube RF amplifier furnishes the necessary excitation.

The heart of the communications system is the Mariner C type transponder. When operated in a coherent mode, where the transmit frequency of the transponder is locked to the frequency of a signal received from the earth in the exact ratio 240/221, determination of spacecraft radial velocity with an accuracy of about 0.02 meter/second is possible. When interrogated by the JPL psuedo noise ranging system, the transponder will permit measurement of the distance to the spacecraft with an accuracy of about ±30 meters.

The receiver portion of the transponder will detect the digital command signal and relay it to the command decoder where it will be temporarily stored until verified on the earth by means of spacecraft telemetry. When correct receipt has been verified, an "execute" signal from the earth will transfer the command to the programmer where it will be acted upon as either a real-time or stored program command.

Figure 13 shows the RF baseband structure for the composite telemetry and video transmission mode, and serves to illustrate the somewhat unusual modulation technique employed for the video data. The 50 bit/second telemetry data train is phase modulated onto a 30-Kc subcarrier, which is then combined with the video data that has been transformed to a vestigial sideband signal. This composite baseband is then phase modulated onto the 2295-Mc RF carrier.

The vestigial sideband signal is created by amplitude modulating the data onto a 310-Kc subcarrier using a double balanced modulator. This suppresses the carrier and produces two equal sidebands. An appropriate filter is then superposed on that double sideband spectrum to essentially eliminate the upper sideband. Since the missing subcarrier must be reinserted on the ground for proper detection of the vestigial sideband signal, provision for deriving such a subcarrier signal is made by transmitting a pilot tone of 38.75 Kc. That pilot tone is exactly 1/8 of the original 310-Kc subcarrier frequency and derived from the same crystal oscillator. Multiplying the received pilot tone by 8 in the ground equipment provides a proper subcarrier for reinsertion. The use of the vestigial sideband modulation technique permits the use of a large modulation index to obtain noise improvement without exceeding the allotted $3\frac{1}{3}$ Mc RF bandwidth.

Reception of the spacecraft data and transmission of commands to the spacecraft will be accomplished using existing NASA Deep Space Receiving Stations. Figure 14 is a block diagram of the equipment used at those receiving stations for handling of the telemetry, video, and tracking data and for generation of commands. Except for the periods when the spacecraft signals are occulted by the moon, the receiving stations located at Goldstone, Woomera, and Madrid will permit continuous contact with the spacecraft. These stations are connected to the NASA Space Flight Operations Facility (SFOF) in Pasadena, which will serve as the central control point for carrying out the mission. The tracking and telemetry data collected at the remote receiving stations are relayed to the SFOF where the necessary computations are made for mission control. Commands to the spacecraft are originated in that facility and relayed to the remote stations for transmission to the spacecraft. Because of the large data bandwidth involved, the video data must be recorded at the sites and the records then shipped to a central processing facility. A computer is used for formatting the telemetry data for transmission to the SFOF and for receiving and storing the spacecraft commands originating at the SFOF.

Analysis of Video System

Figure 15 shows all of the elements which make up the video system, both in the spacecraft and on the ground. Defining the overall transfer function of that system is accomplished by cascading the input-output transfer functions of each of the system elements. The system output, which consists of the system excitation (the scene) times the transfer function can then be compared with the system noise, passed through the system elements, in order to obtain the overall signal to noise ratio. Figure 15 is a normalized plot of the spatial frequency response of each of the system elements. Nominal conditions such as lens off-axis response and variation of film response with off-nominal density are not shown, but can be included in the analysis if desired. The "smear" transfer function is shown for the worst case computed exposure of 1/25 second and 2σ variation in control capability in vehicle pointing and rate. Halving the exposure time would effectively increase this transfer function to within 20 percent of unity in the range shown. The overall system transfer function is shown by the heavy curve.

The primary sources of system noise are the spacecraft recording film and the communications link, although minor noise inputs are inserted by other elements. When the system noise is combined with the system transfer function the overall signal to noise ratio is as shown in figure 17. The system test input is considered to be the bar chart test pattern shown. The intensity of the alternate bars coincides with the intensity of light which could exist on the two meter base, 1/2-meter high cone (an assumed lunar obstruction of importance to the Apollo lunar excursion module). Similar lighting would prevail for a lunar crater of the same dimensions. If a bar target having bars of 1/2 meter in width were scanned by an optical aperture of effective 1/2-meter diameter, a signal to noise of about 0.7 would be obtained. With the target bars 1 meter in size, a signal to noise of six could be secured. For craters or cones of the size indicated, the computed signal to noise would be approximately 3. This is considered sufficient for adequate target detection and provides a sufficiently low rate of false target generation. It is probable that trained film interpreters could exceed the computed system performance for target detection using integration schemes which are at present not amenable to numerical evaluation.

Concluding Remarks

The Lunar Orbiter system has been designed and appears adequate to secure the photographic coverage required for the Apollo Project. At the present time, almost all designs have been completed and fabrication is underway. Starting this summer an intensive series of tests will commence on all subsystems and components, to be followed by spacecraft level testing lasting through 1965. First launchings of the Orbiter are planned for 1966.

Reference

1. Taback, I.: Lunar Orbiter: Its Mission and Capability. Paper presented at American Astronautical Society Meeting, May 4-7, 1964.

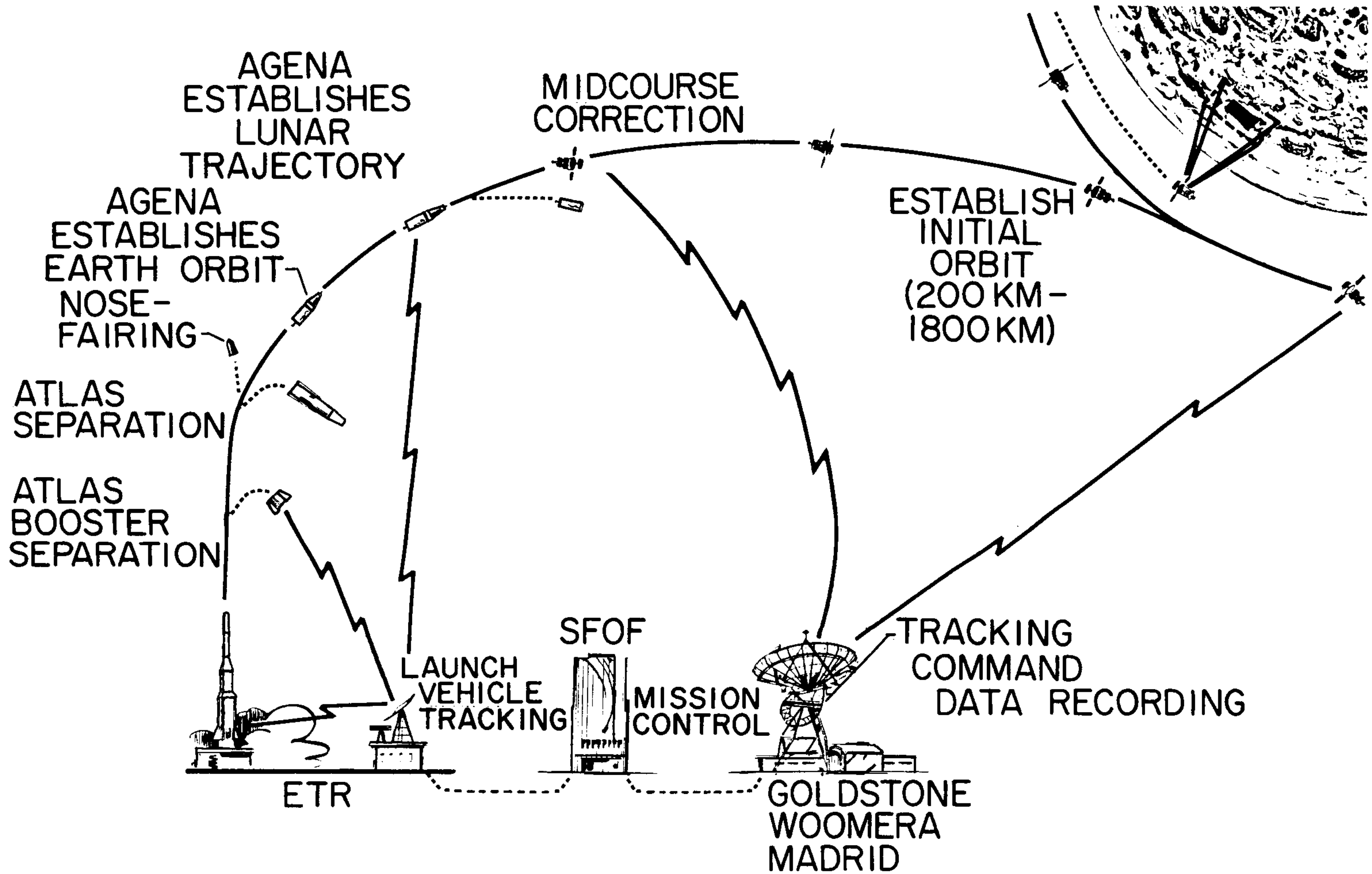

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Figure 1.- Mission operations.

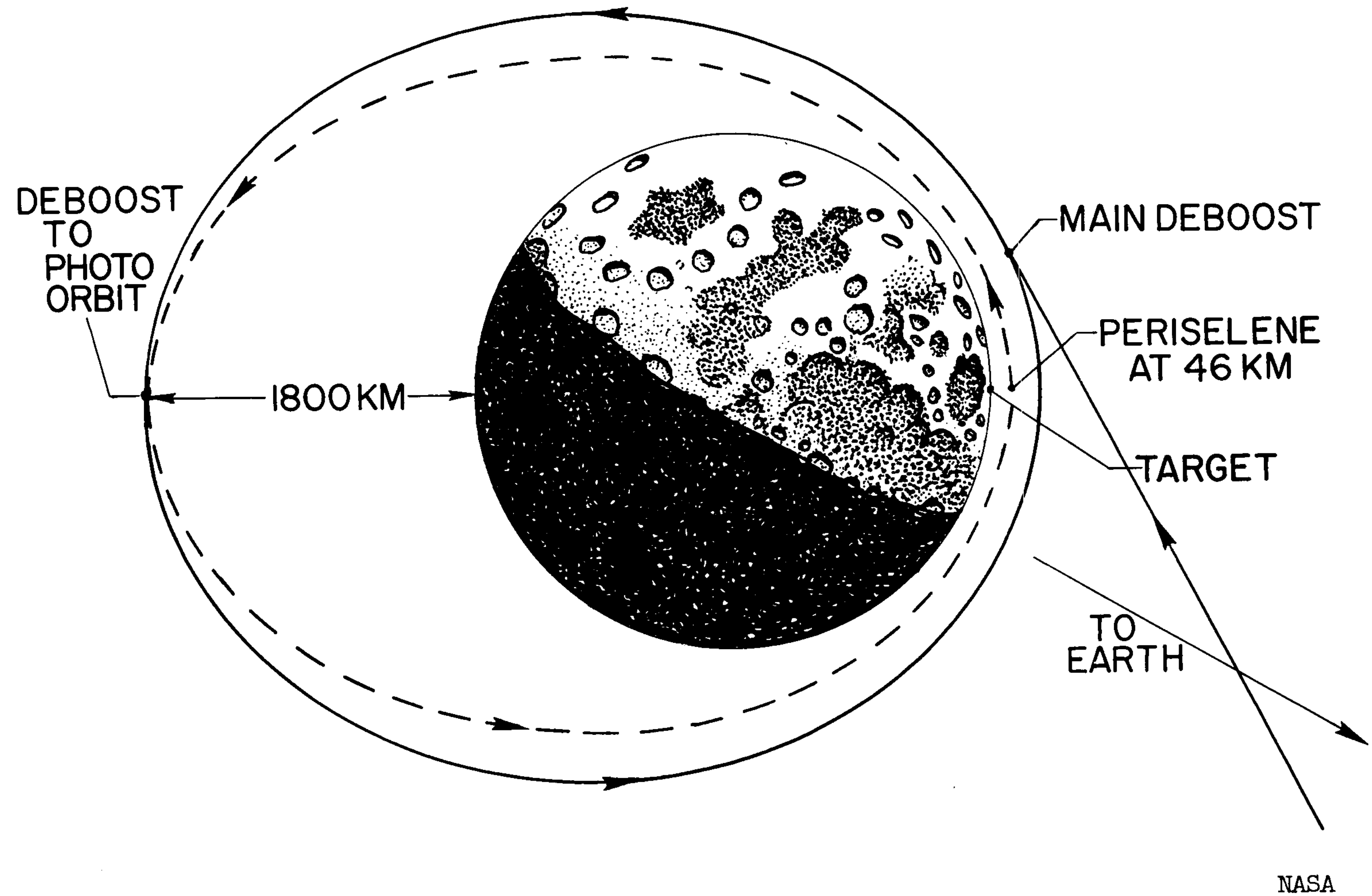

Figure 2.- Lunar orbital maneuvers shown in approach hyperbola plane.

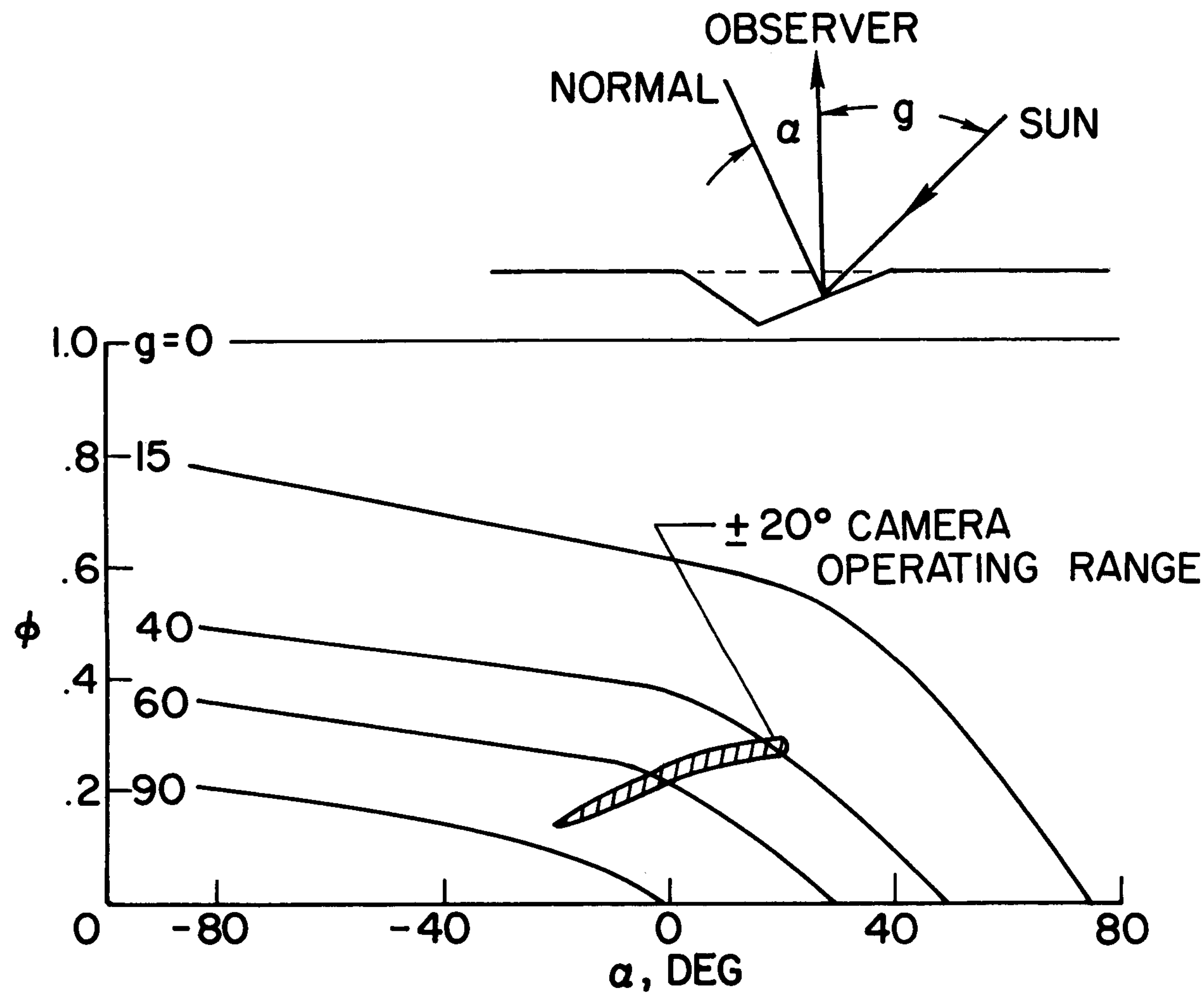

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Figure 3.- Lunar photometric function.

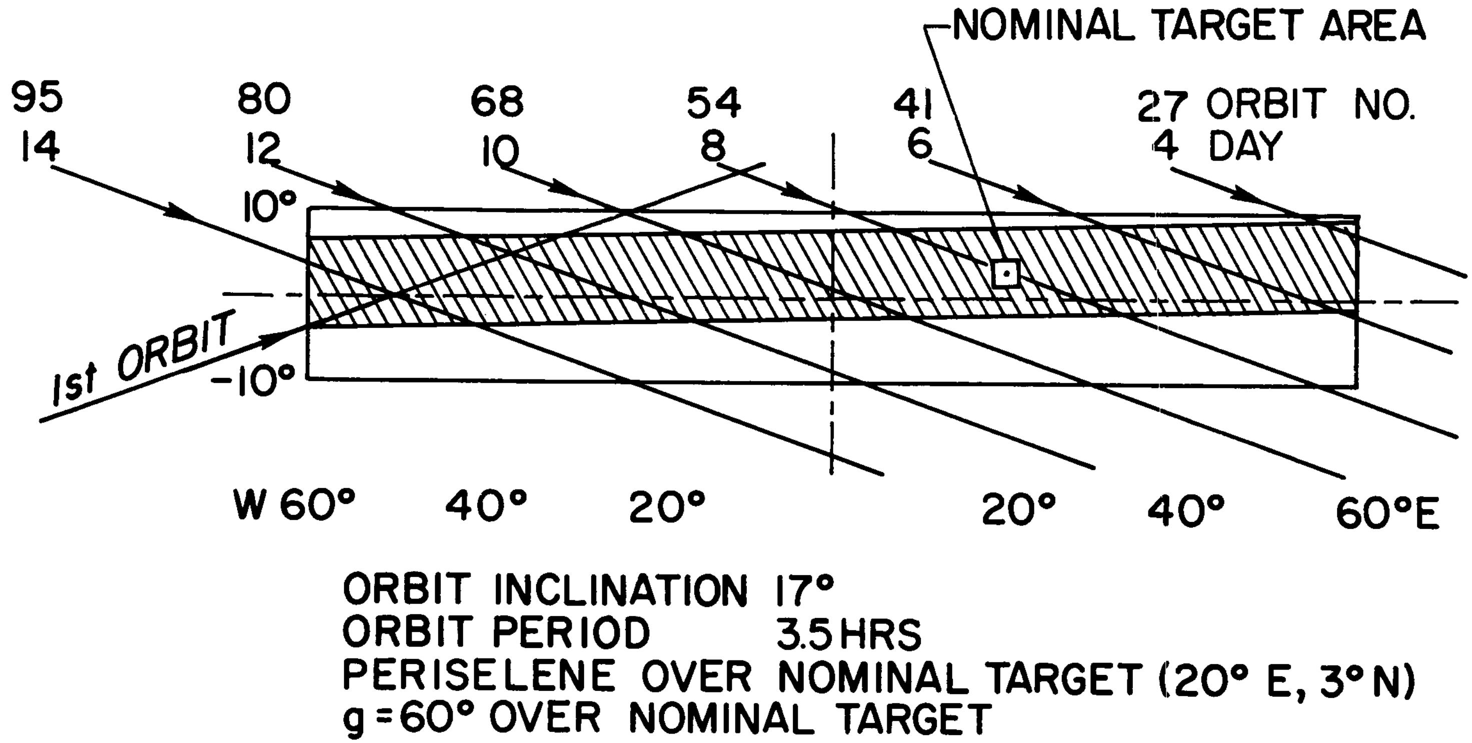

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Figure 4.- Typical target selection area.

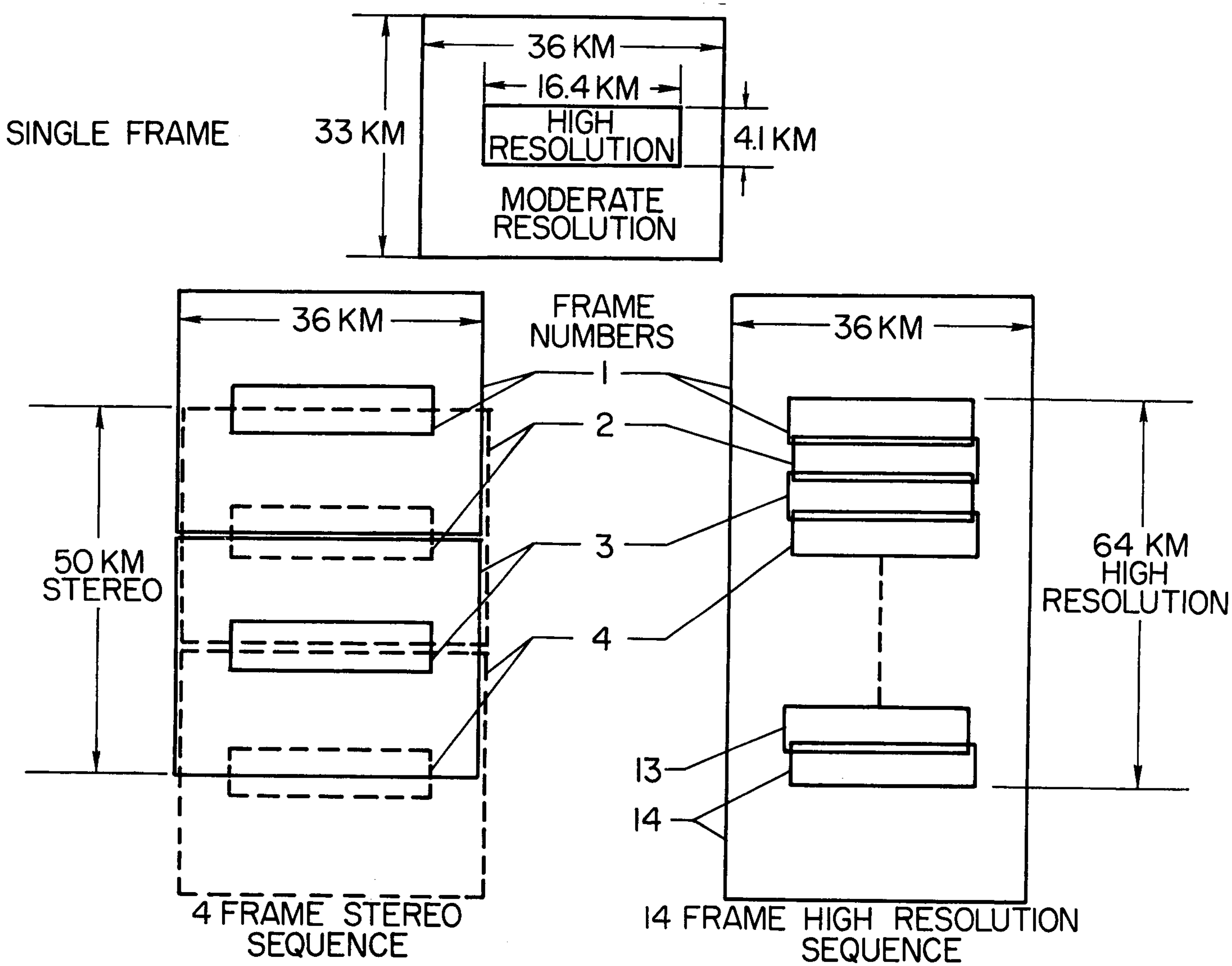

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Figure 5.- Frame format and typical sequences.

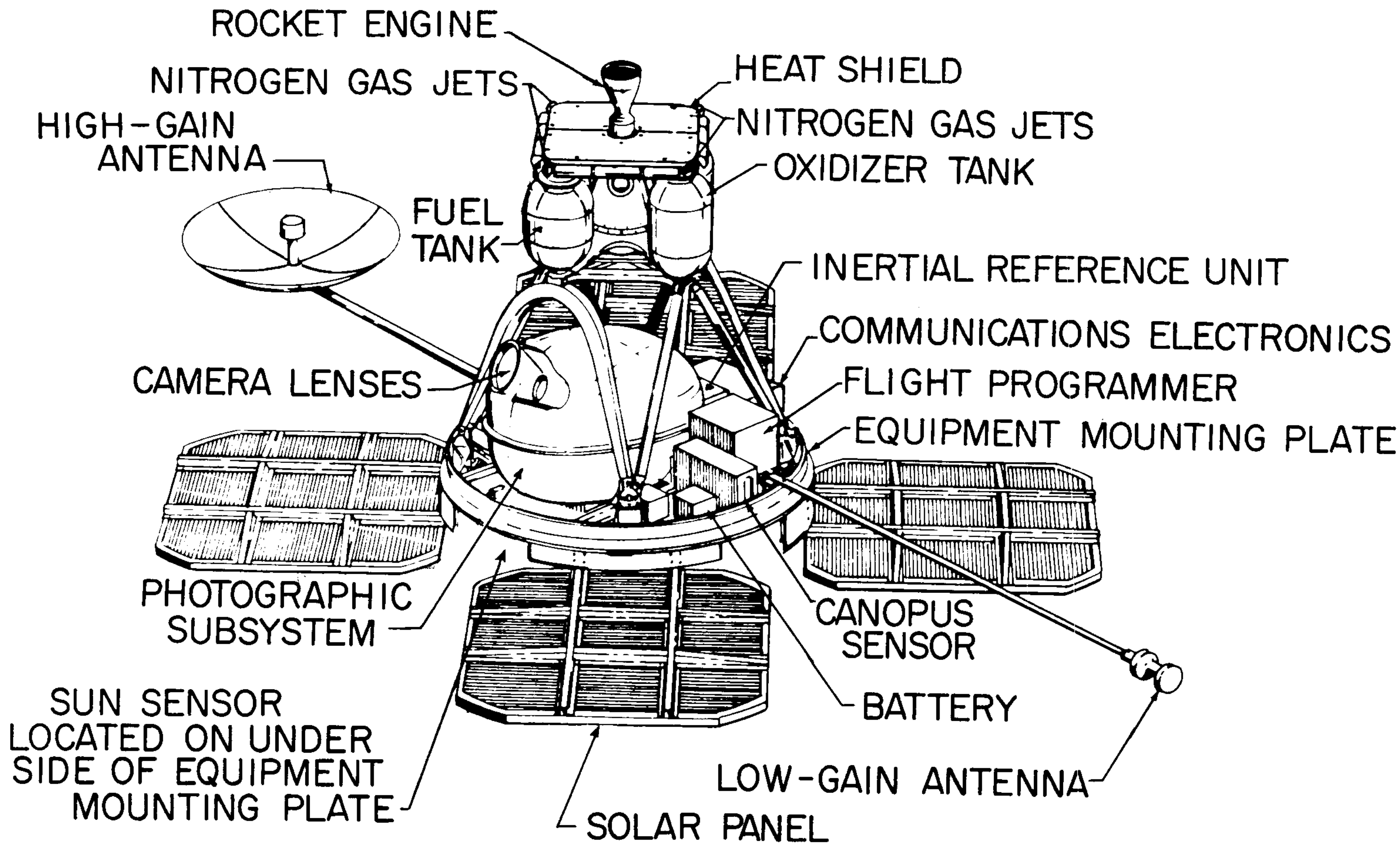

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Figure 6.- Spacecraft configuration.

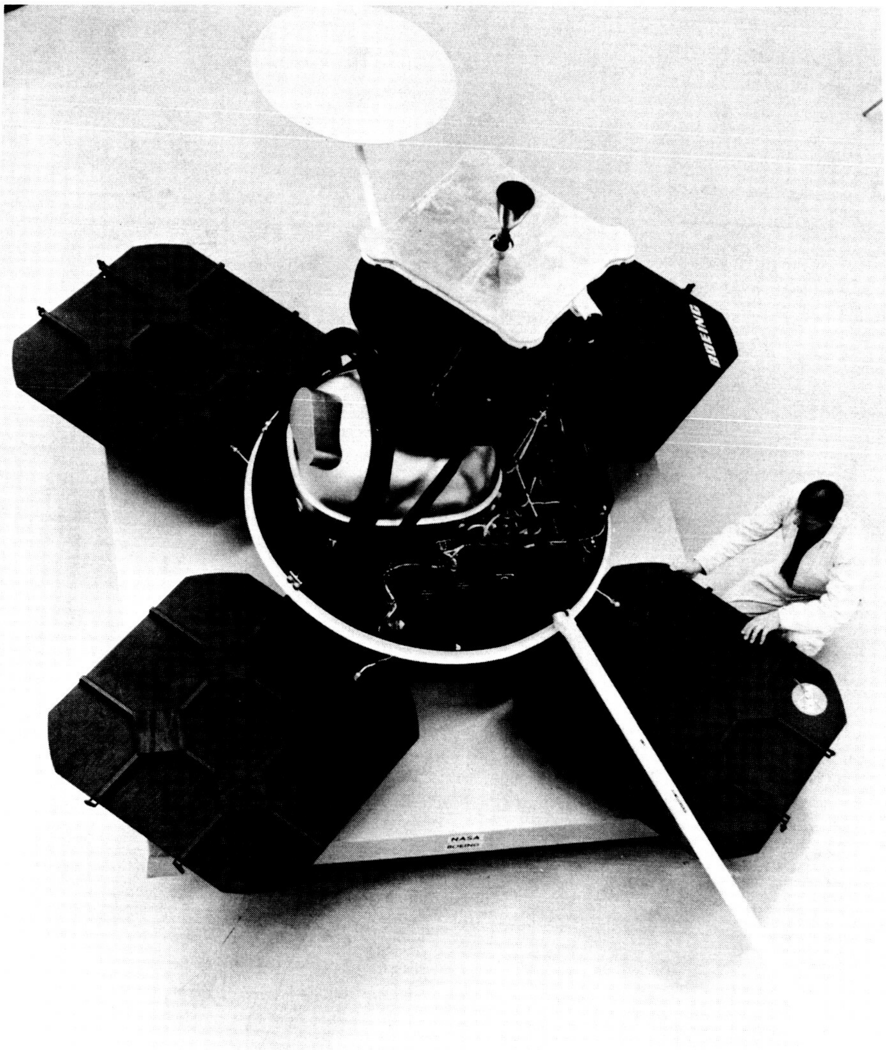

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Figure 7.- Lunar orbiter mockup.

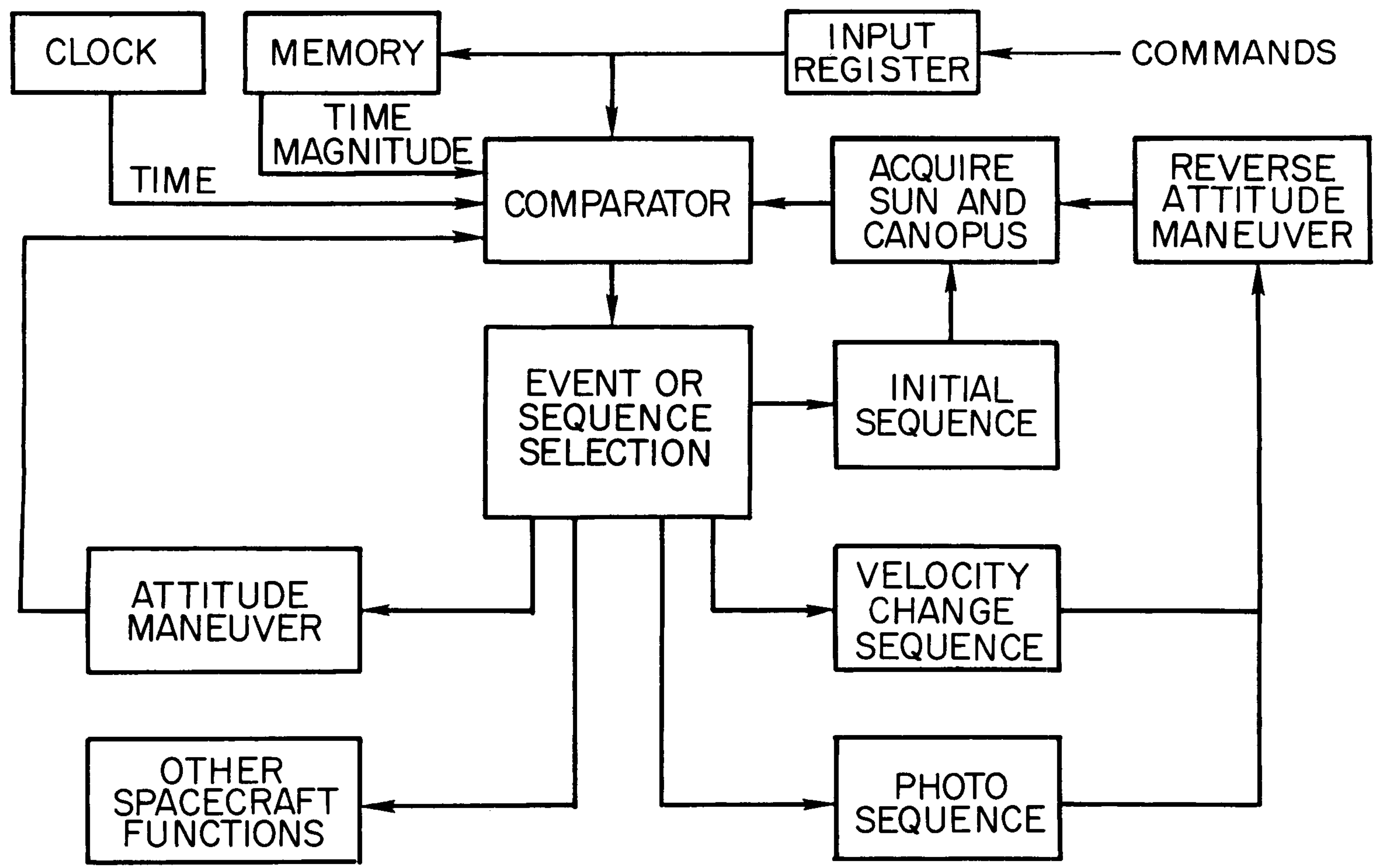

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Figure 8.- Programmer flow diagram.

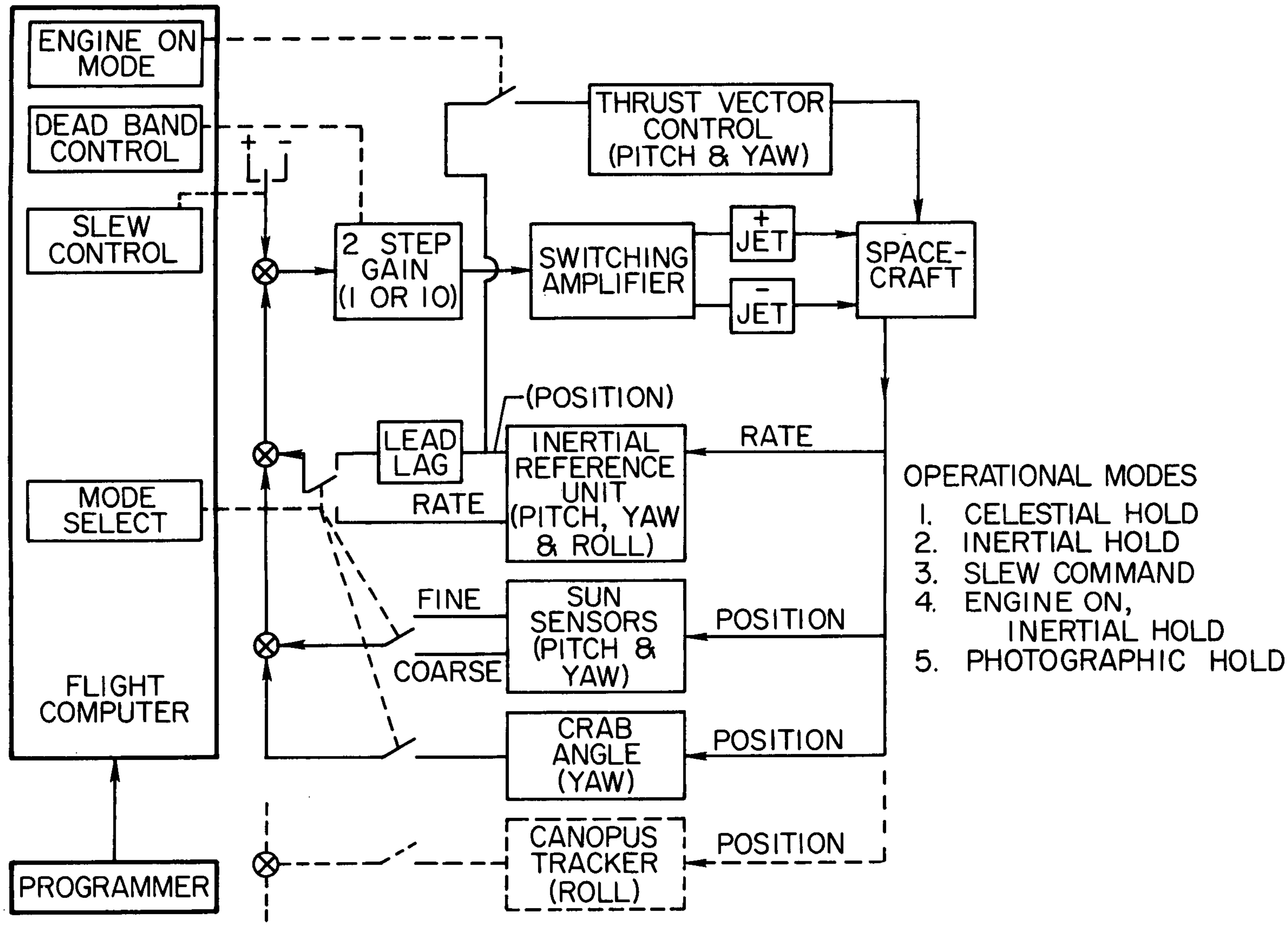

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Figure 9.- Attitude control system (yaw channel).

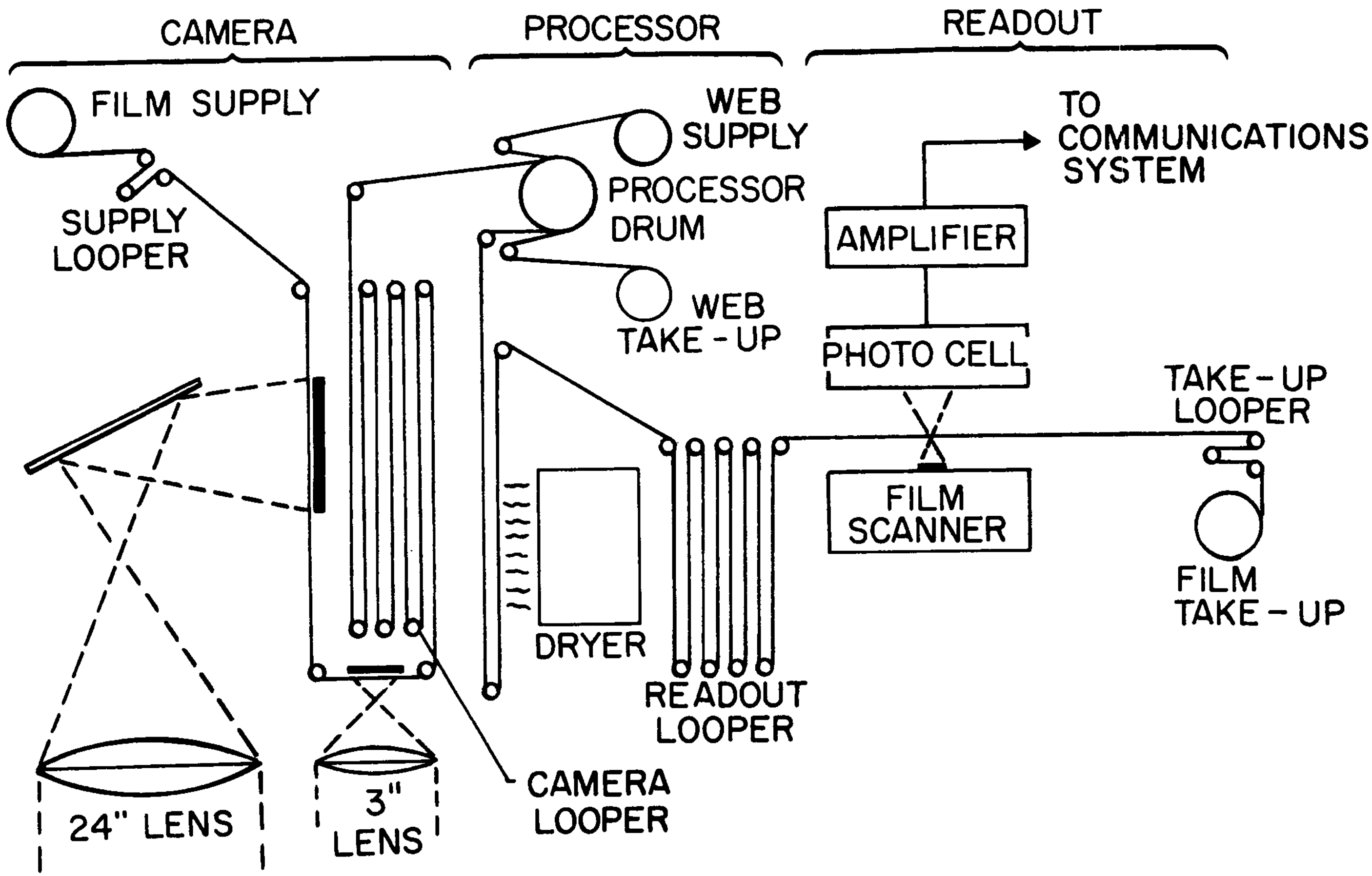

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Figure 10.- Spacecraft photographic system.

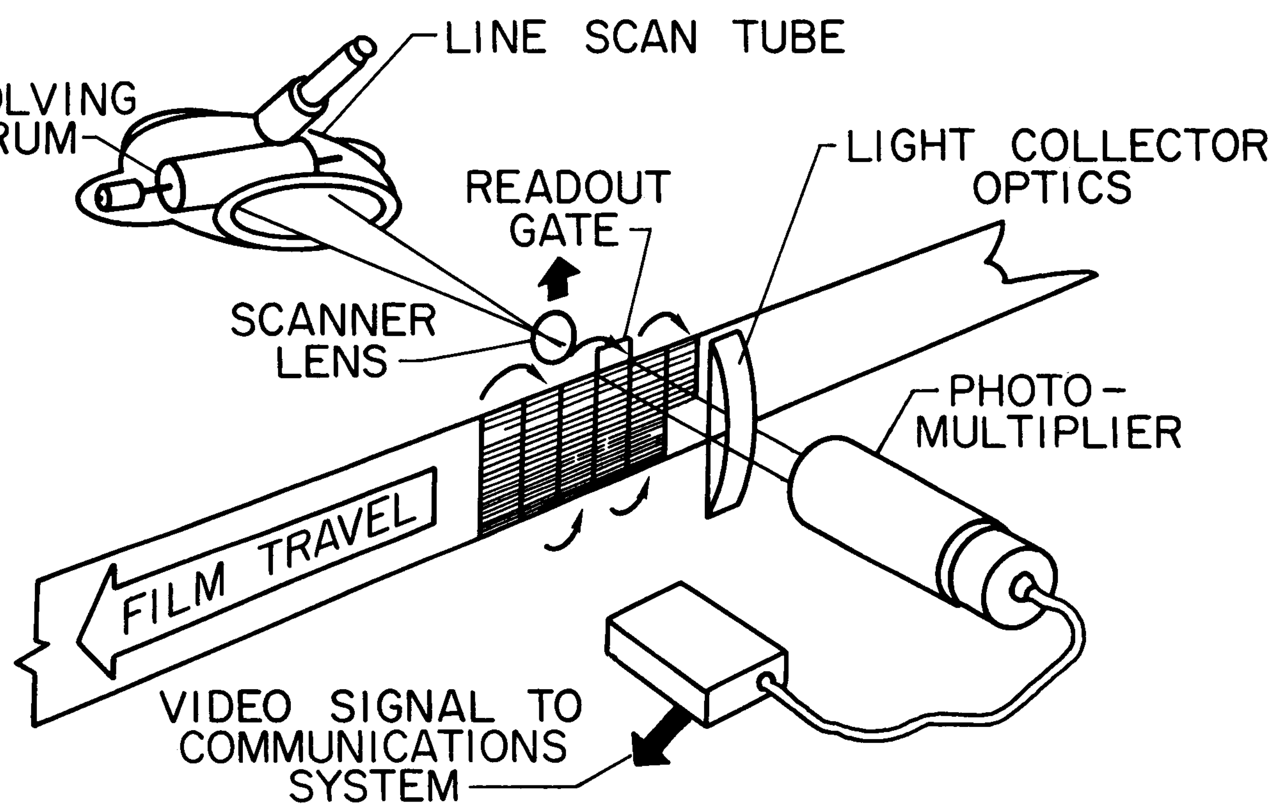

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Figure 11.- Photographic system readout schematic.

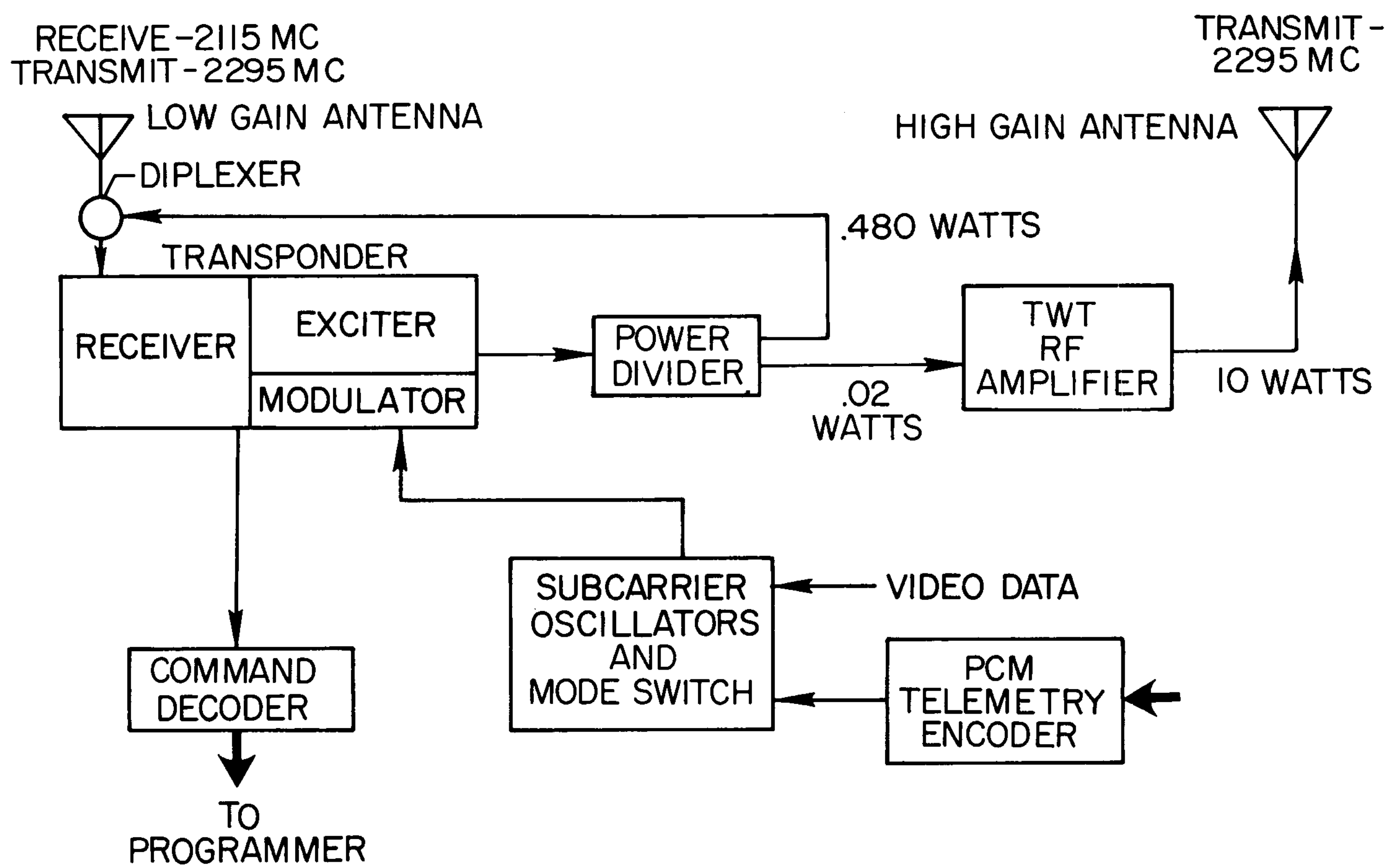

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Figure 12.- Spacecraft communications system.

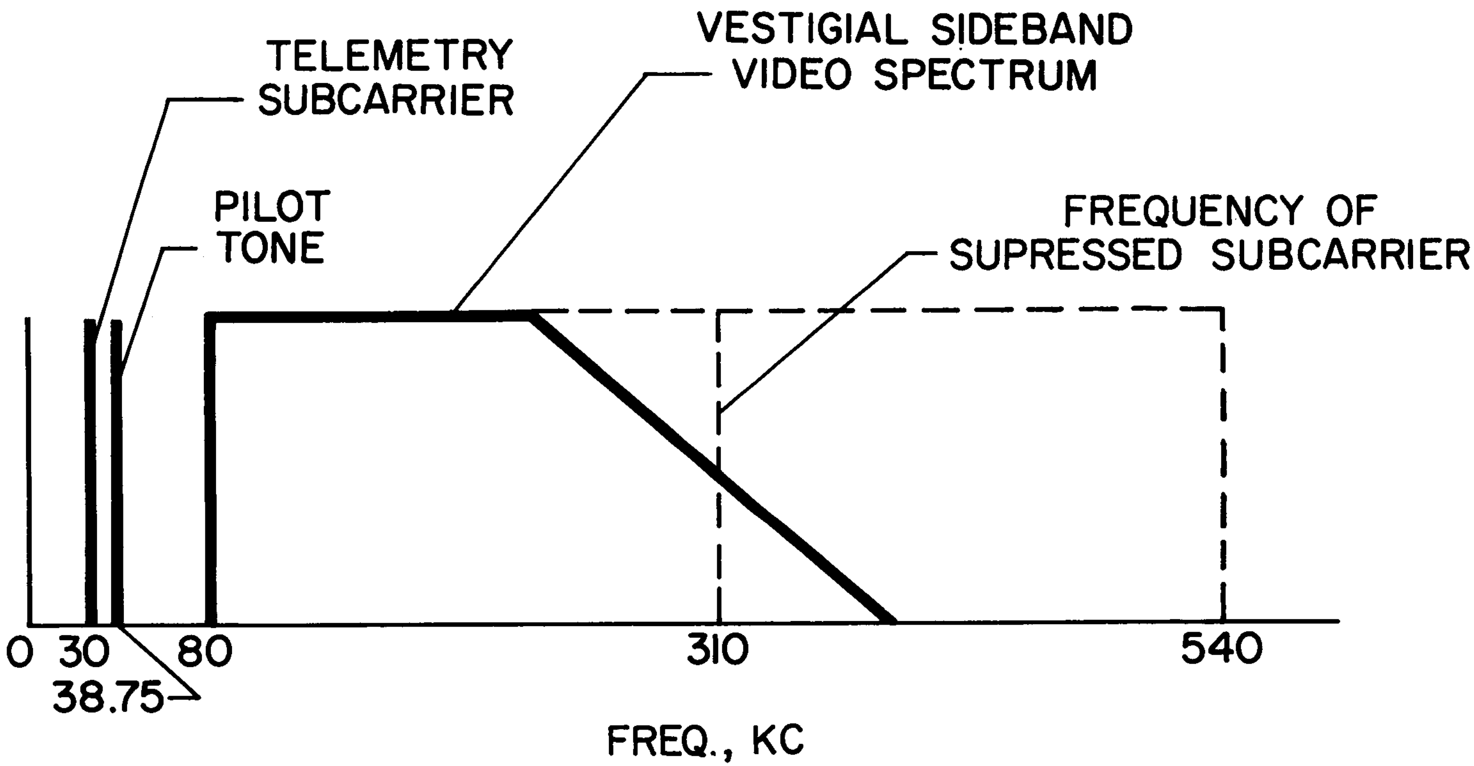

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Figure 13.- RF baseband structure.

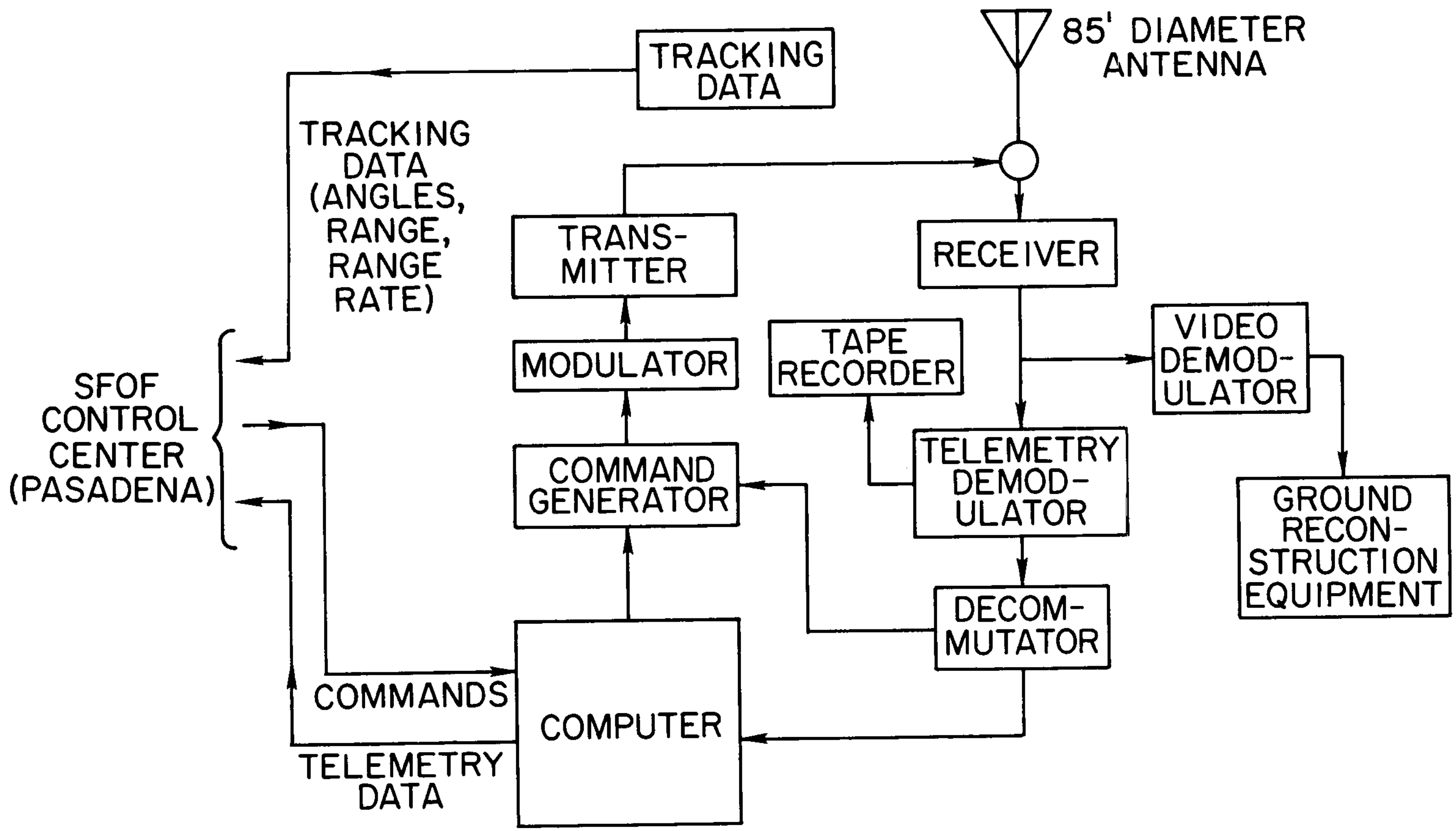

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Figure 14.- Typical receiving station.

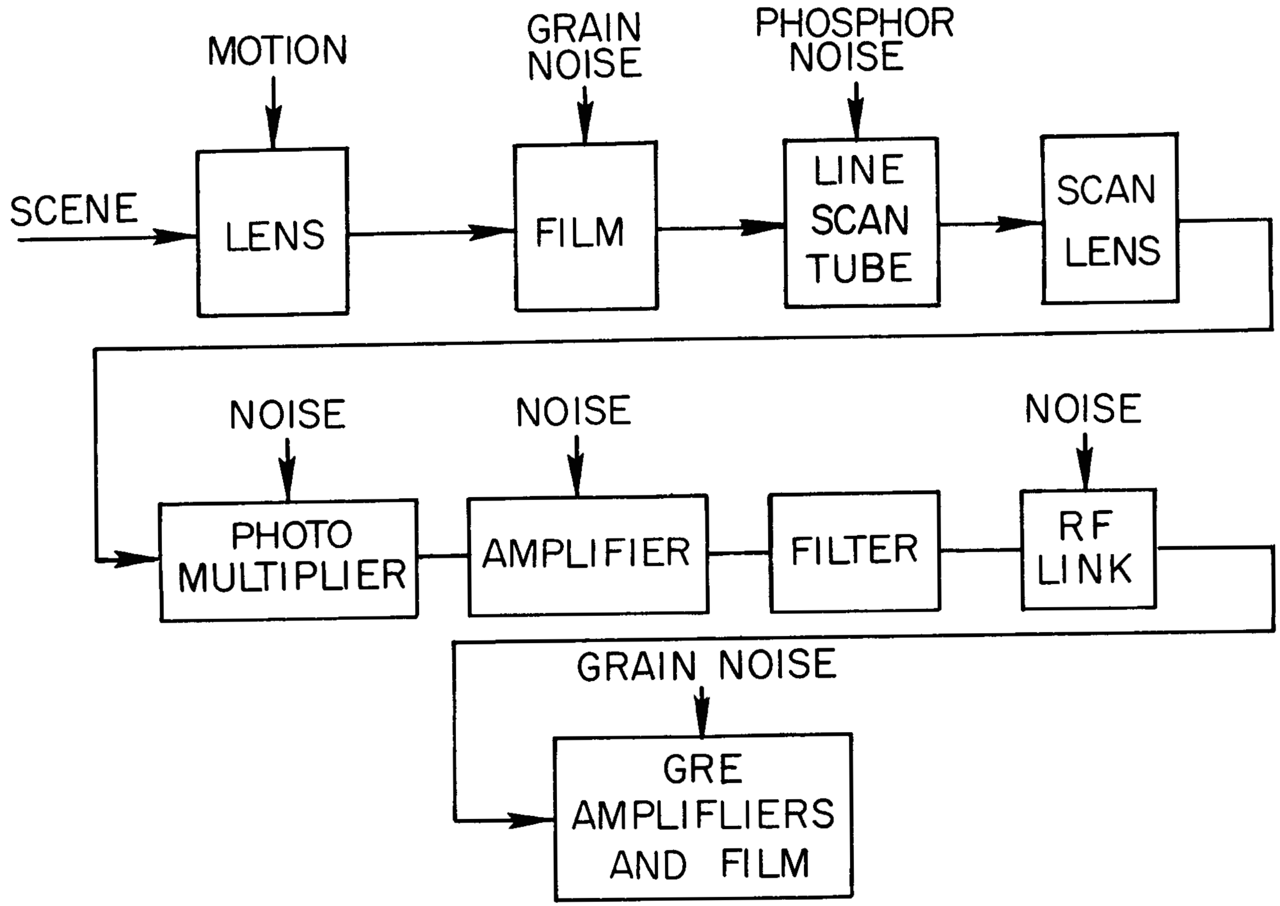

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Figure 15.- Video system elements.

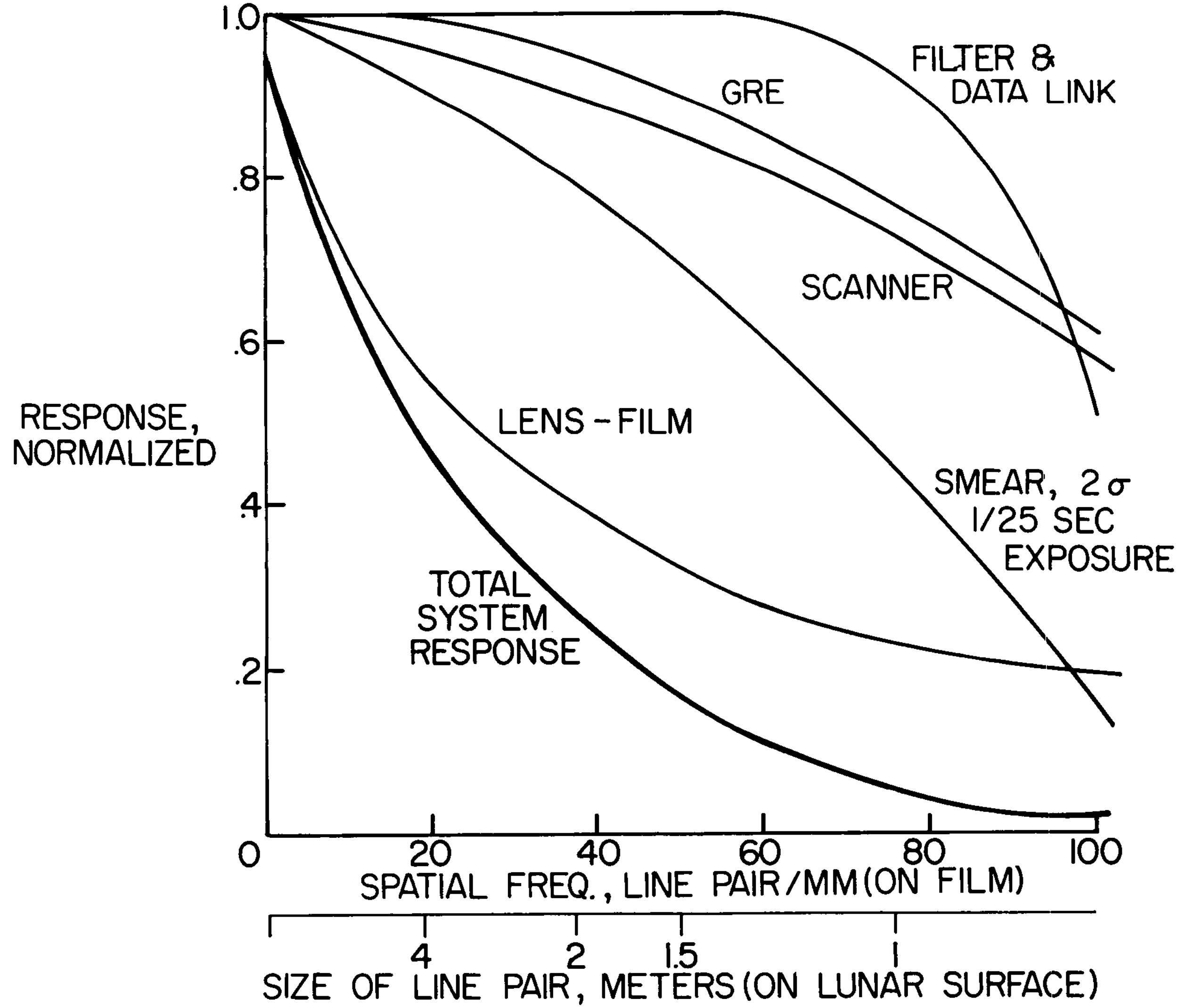

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Figure 16.- Response of video system elements.

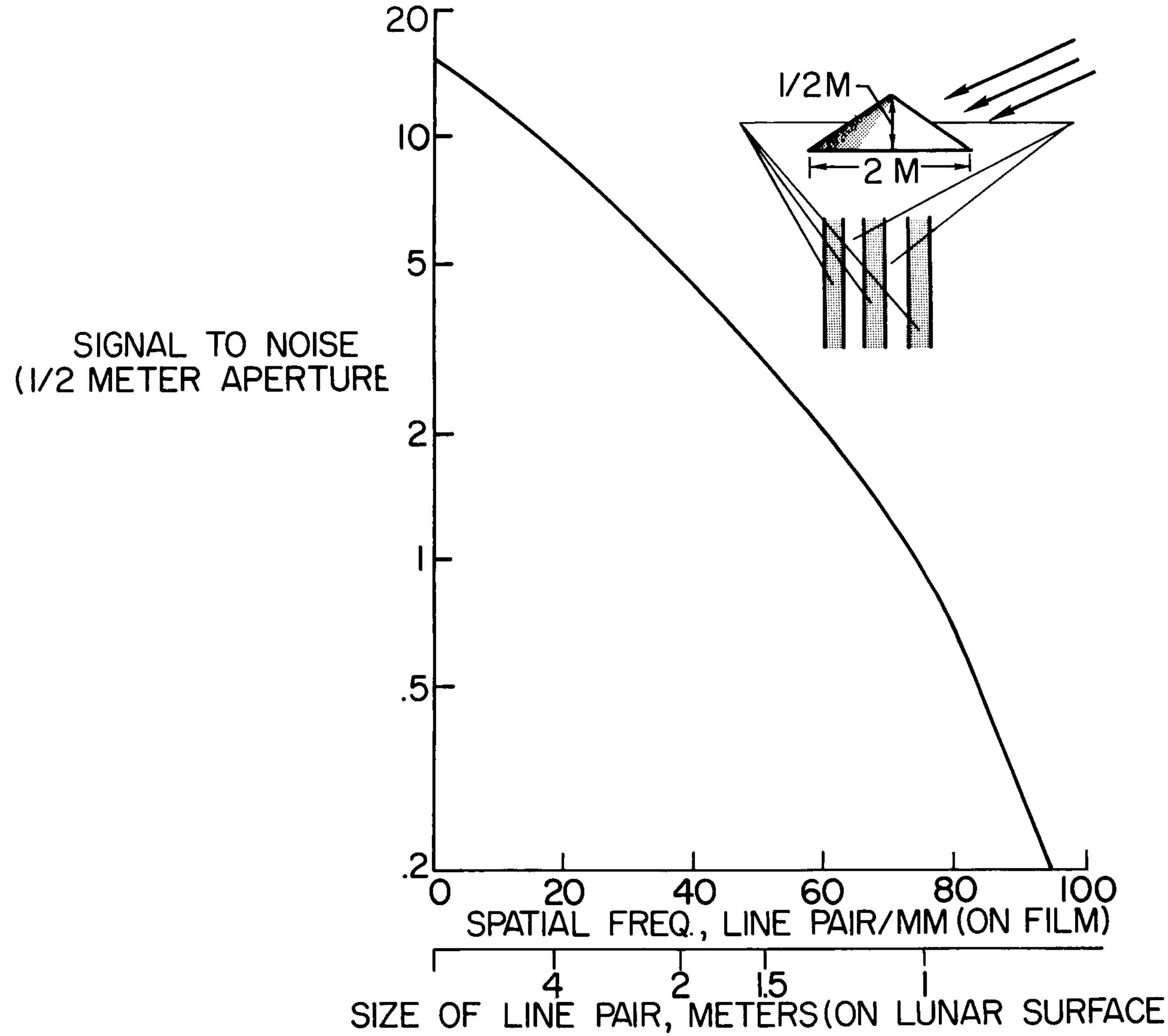

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Figure 17.- System signal to noise ratio.

NASA-Langley, 1965